N71-20501

NASA TECHNICAL MEMORANDUM

NASA TM X-52980

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AN IMPROVED METHOD FOR MAXIMIZING THE PAYLOAD OF ELECTRIC PROPULSION SPACECRAFT AT LOW POWER LEVEL

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February, 1971

This information is being published in preliminary form in order to expedite its early release.

E-6196

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ABSTRACT

Installing less than the optimum amount of power on an electrically propelled spacecraft can drastically reduce its payload capability. A technique for improving the payload at low power is proposed which involves reducing the total mass of the spacecraft. It is shown that an optimum total mass of the electric propulsion spacecraft can be less than the maximum injected mass capability of the chemical-rocket launch vehicle at any given injection velocity. An example of the method is given for a Mercury orbiter mission using a solar-electric-propulsion spacecraft launched by a Titan IIID/Centaur.

AN IMPROVED METHOD FOR MAXIMIZING THE PAYLOAD OF ELECTRIC PROPULSION SPACECRAFT AT LOW POWER LEVEL

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SUMMARY

A method is described which, for a given combination of launch vehicle and electric propulsion spacecraft, improves gross payload capability at low electric propulsion system power levels. An optimized design of high electric propulsion system power level and corresponding high payload is used as a reference point. For lower power, the payload and other spacecraft masses are scaled down from the reference point in direct proportion to the ratio of desired to reference power level. The initial mass of the electric propulsion spacecraft is hence scaled down by the same factor, requiring that the full injection mass capability of the launch vehicle not be used. Nevertheless, this approach often yields higher payloads at sub-optimum power level than alternative methods in which the spacecraft initial mass always reflects full use of the launch vehicle. The scaling method can be used with or without constraints on electric thruster specific impulse. An example and discussion of the method is given for a Mercury orbiter mission using a solar-electric propulsion spacecraft launched by a Titan IIID/Centaur.

INTRODUCTION

The "boosted" electric propulsion mission mode, where a launch vehicle places the electric spacecraft on an earth escape path at the start of each mission, has been widely used to evaluate multi-mission applications of specific combinations of a launch vehicle and a fixed-power-level electric propulsion system. For example references 1, 2, and 3 suggest multi-mission applications of a 10 kW solar-electric propulsion system with either an Atlas/Centaur or Titan IIIC launch vehicle. Often, the propulsion system power levels of interest are sub-optimum (less than the optimum for maximum payload) since, especially in solar-electric systems, system cost can be assumed to be directly proportional to power.

Recently, large and relatively cost-effective launch vehicles, such as Titan IIID/Centaur, have been proposed for the launching of electric propulsion spacecraft. Larger launch vehicles, however, often call for high power level electric propulsion systems for an optimum combination of launch vehicle and electric propulsion spacecraft. Therefore, efforts to hold the electric propulsion system power (and cost) at the same levels used with smaller launch vehicles require operation at much lower power levels relative to the optimum. However, attempting to achieve low power operation relative to the optimum level can result in disappointing loss of performance of the electric propulsion spacecraft. At the same time, the optimum specific impulse of the electric propulsion system may be undesirably low, corresponding to highly inefficient and, perhaps unreliable, operation.

This memorandum proposes a simple scaling technique[1] which can, in many cases, greatly improve the payload capability at sub-optimum power levels for any combination of launch vehicle and electric propulsion spacecraft. Furthermore, the method can be used with or without constraints on the specific impulse of the electric thrusters. The scaling procedure results in the maximum possible payload for any specified combination of specific impulse and sub-optimum power level. It can therefore have an important effect in two critical areas of electric propulsion mission analysis. These are: (1) achieving the best possible payload from a combination of a very large launch vehicle and an electric propulsion system of limited size, and (2) as an aid to better specification of the power and specific impulse of the best multi-mission electric propulsion system.

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[1]As this paper was being prepared for publication, it was learned that the basic concept had been recognized, although not further discussed, by TRW Systems Group in their document no. 16552-6003-TO-00 dated January 15, 1971, the second quarterly progress report under NASA Contract NAS2-6040.

SYMBOLS

I_{sp}	specific impulse, sec
m_{BR}	braking rocket system mass, kg
m_{NS}	net spacecraft mass, kg
m_{oe}	total mass of spacecraft, kg
m_p	propellant mass, kg
m_{ps}	electric propulsion system mass, kg
m_t	tankage mass, kg
P	propulsion system power, kW
P_T	power at point of tangency, kW
V_B	characteristic injection velocity, km/sec
α	electric propulsion system specific mass, kg/kW
η	overall efficiency of ion thrusters

EXAMPLE AND DISCUSSION

Net Spacecraft Mass

As defined in reference 4, the net spacecraft mass, m_{NS}, is simply the mass remaining after the ion propellant and tankage, m_p and m_t, the electric propulsion system, m_{ps}, and retro-braking rocket system, m_{BR}, (if any) are subtracted from the initial mass of the electric propulsion spacecraft, m_{oe}.

$$m_{NS} = m_{oe} - m_p - m_t - m_{ps} - m_{BR}$$

The net spacecraft mass is therefore a gross payload parameter and is maximized by finding optimum combinations of m_p, m_{ps}, m_{BR}, and m_{oe} for each given mission.

Initial Mass

For "boosted" electric propulsion missions, the injection velocity, V_B, is assumed to be equal to or greater than earth escape speed at low altitude - about 11 km/sec. The electric propulsion spacecraft initial mass, m_{oe}, is then assumed to vary with V_B along a launch vehicle capability curve, such as is shown in figure 1. Hence, due to its effect on m_{oe}, V_B is introduced as an additional parameter in the maximization of m_{NS}.

Figure 1 shows the effect of V_B on injected mass capability for the Atlas/Centaur and the Titan IIID/Centaur.

Sub-Optimum Power

Values of m_{NS} for boosted solar-electric propulsion systems using each of the two launch vehicles of figure 1 can be compared in figure 2. Figure 2 shows a typical variation of m_{NS} with propulsion system power level for a Mercury orbiter mission with a 500-day trip time. For these cases the specific mass, α, of the solar-electric propulsion system was assumed to be 30 kg/kW at 1 AU. Power output of the solar-electric system varies with distance from the Sun as discussed in reference 1. Overall efficiency, η, of the ion thruster system varies with thruster specific impulse, I_{sp}, as discussed in reference 2. The Mercury parking orbit for this example has been arbitrarily chosen as an ellipse with a periapsis of 6 Mercury radii and an eccentricity of 0.5.

The solid curves in figure 2 are envelope curves of optimized I_{sp} and V_B for maximum m_{NS} at each power level. The dashed curve underlying the envelope curve in figure 2 is included to illustrate (for this mission) typical data for constant I_{sp}.

Figure 2 illustrates a major problem that may accompany the use of a large launch vehicle. For the Titan IIID/Centaur launch vehicle, the optimum power level for the electric propulsion spacecraft is about 50 kW. Reducing the spacecraft power level to below 20 kW requires an excursion far off the optimum, causing a major penalty in m_{NS}.

Figure 3 shows the optimum values of I_{sp} and V_B corresponding to various power levels for the Titan IIID/Centaur/solar-electric example in figure 2. In general, lower power levels are made possible by decreasing the I_{sp} of the ion thruster system. If, as in this example, a boosted mission mode is used, both V_B and I_{sp} can be re-optimized as power is reduced. Such cases show the increase in optimum V_B and decrease in optimum I_{sp} with decreasing power seen in figure 3. A large excursion down from optimum power levels can call for very low I_{sp}; e.g., 1500 seconds. Electrostatic thrusters are relatively inefficient at such low I_{sp}, and it is also doubtful whether current technology could make them operate reliably.

One obvious alternative for better performance at very low power would be to drop back to the use of the smaller launch vehicle, as indicated by the Atlas/Centaur/solar-electric data in figure 2. In this case, only extremely low power levels (below 4 kW) would require an I_{sp} below 2000 seconds. However, this change in launch vehicle could result in too large a decrease in payload capability with no allowance for potential improvement. For example, in figure 2 the system launched by Atlas/Centaur achieves an overall maximum m_{NS} of 450 kg at a power of 11 kW.

Such a step to a smaller launch vehicle for improved performance at very low powers may be desirable - but is not necessary. The scaling method described in the following section of this memorandum allows better utilization of the large launch vehicle, such as Titan IIID/Centaur, in the low power range resulting in values of m_{NS} that equal or exceed those possible with the smaller launch vehicle.

Scaling Method

The scaling method depends on "off-loading" the launch vehicle, for want of a better term. That is, the launch vehicle capability curve shown in figure 1 is re-interpreted as showing only the upper limit of m_{oe} at each V_B, not necessarily the most advantageous value. It is assumed that at each V_B the electric propulsion spacecraft would be scaled down in power, payload, and initial mass by a common factor. Under certain conditions (defined in the following section) the resultant value of m_{NS} for low power level (obtained by scaling) is greater than would be found at the same fixed power from the envelope curve of figure 2. The basic steps in this procedure are illustrated in figure 4 which repeats the envelope curve of m_{NS} with P given in figure 2.

First, a tangent line is drawn from the origin to the envelope curve of m_{NS} for the given combination of launch vehicle and electric spacecraft. The tangency point identifies P_T, the power level at which the ratio m_{NS}/P is a maximum for the given envelope curve. The tangent line is then the locus of maximum possible m_{NS} for all powers less than P_T, but only for electric propulsion spacecraft which are scaled-down versions of the spacecraft designed for the power P_T. As power is decreased, injection velocity, V_B, and I_{sp} remain constant at the optimum values corresponding to P_T. Along the tangent line the optimum electric propulsion spacecraft design is linearly scaled in m_{NS} and m_{oe} with power.

In the case illustrated in figure 4, the point of tangency occurs at power level of 27 kW. Referring to figure 3, the optimum V_B and I_{sp} for 27 kW are 11.12 km/sec and 2600 seconds respectively. For a power level of 13.5 kW ($P/P_T = 0.5$), the optimum values of V_B and I_{sp} remain the same, but the best m_{oe} for the electric spacecraft is half the launch vehicle capability at 11.12 km/sec. In this example the m_{NS} of about 800 kg is also half the value possible for $P/P_T = 1$. But this m_{NS} is far greater than that possible for the non-scaled spacecraft ease and, in fact, is the greatest possible value of m_{NS} for this choice of mission, power level, and launch vehicle.

For clarity, the scaling method has been described here in graphical terms. In actual practice, the scaling procedure needs only the identification of the point of maximum m_{NS}/P. A calculation procedure for maximizing m_{NS}/P should be used. For all power levels below the P corresponding to the point of maximum m_{NS}/P, the payload and other spacecraft masses are scaled down linearly with power. However, for higher power levels scaling does not apply and the envelope curve of optimized I_{sp} and V_B for maximum m_{NS} must be calculated.

Multiple Launching

It should be noted that more than one low power electric spacecraft could be launched by the launch vehicle rather than discarding large fractions of the launch capability at the optimum V_B. For example when P/P_T is 0.50 or 0.33, ideally, two or three identical electric spacecraft could be launched, each having the same payload capacity and power level. Multiple launching of dissimilar size spacecraft is also possible, each being scaled versions of the other, as long as the total injected mass capability of the launch vehicle at the given V_B is not exceeded. The desirability of such multiple launch approaches of course depends on the need for multiple spacecraft on a given mission, or the possibility that spacecraft for two different missions could share a common launch window and injection velocity.

Limitations

This technique relies on the capability of scaling the electric propulsion spacecraft in m_{oe} and P for proportionate changes in m_{NS}. In general, such scaling is valid in electric propulsion spacecraft as long as fixed mass penalties in the various spacecraft subsystems, such as the powerplant, are relatively small. If such fixed mass penalties are significant, the true curve of maximum m_{NS} would drop lower than the tangent line in figure 4 as lower powers (and lower m_{oe}) are employed.

There are combinations of mission, launch vehicle, and electric propulsion system parameters for which scaling techniques are not applicable. Re-examination of figure 4 shows that a tangent line can be drawn to the performance curve (for either constrained or unconstrained I_{sp}) only when m_{NS} equals zero for some positive value of P. When P is equal to zero, an m_{NS} of zero or greater indicates a mission which could be accomplished by the launch vehicle alone, even thoughan additional electric propulsion system may be beneficial.

Constrained I_{sp}

Scaling the electric propulsion spacecraft along the tangent line from the point of maximum m_{NS}/P yields the best possible m_{NS} for all powers less than P_T. But the optimum I_{sp} at the tangent point may be less than desired for operation of the ion thruster system. In such cases, the basic scaling procedure is simply modified to create a new scale line for the best possible m_{NS} for constrained I_{sp}.

Referring back to figure 2, a typical data curve is shown for I_{sp} held constant at 3000 seconds. This curve could serve as a reference for constructing a new tangent line from the origin for scaled spacecraft having I_{sp} fixed at 3000 seconds. In other words it is only necessary to calculate the maximum m_{NS}/P with I_{sp} constrained, thus identifying the corresponding P and m_{NS}. All previous comments about scaled electric spacecraft designs apply to the use of the new reference point, although new values of I_{sp}, V_B, and P_T are used. In addition, the new values of m_{NS} at each power will be less than those given by the tangentline to the envelope curve for unconstrained I_{sp}.

CONCLUSION

It has been shown that an optimum design procedure for sub-optimum power electric propulsion spacecraft should consist simply of seeking the maximum of the ratio m_{NS}/P for each combination of launch vehicle and mission. Various maximum m_{NS}/P ratios can be found with or without constraints on I_{sp}. The electric propulsion spacecraft with maximum m_{NS}/P, or a scaled-down version of it, results in the most effective utilization of a given launch vehicle as electric propulsion system power is decreased.

Lewis Research Center
National Aeronautics and Space Administration
Cleveland, Ohio, February 3, 1971
124-08-41

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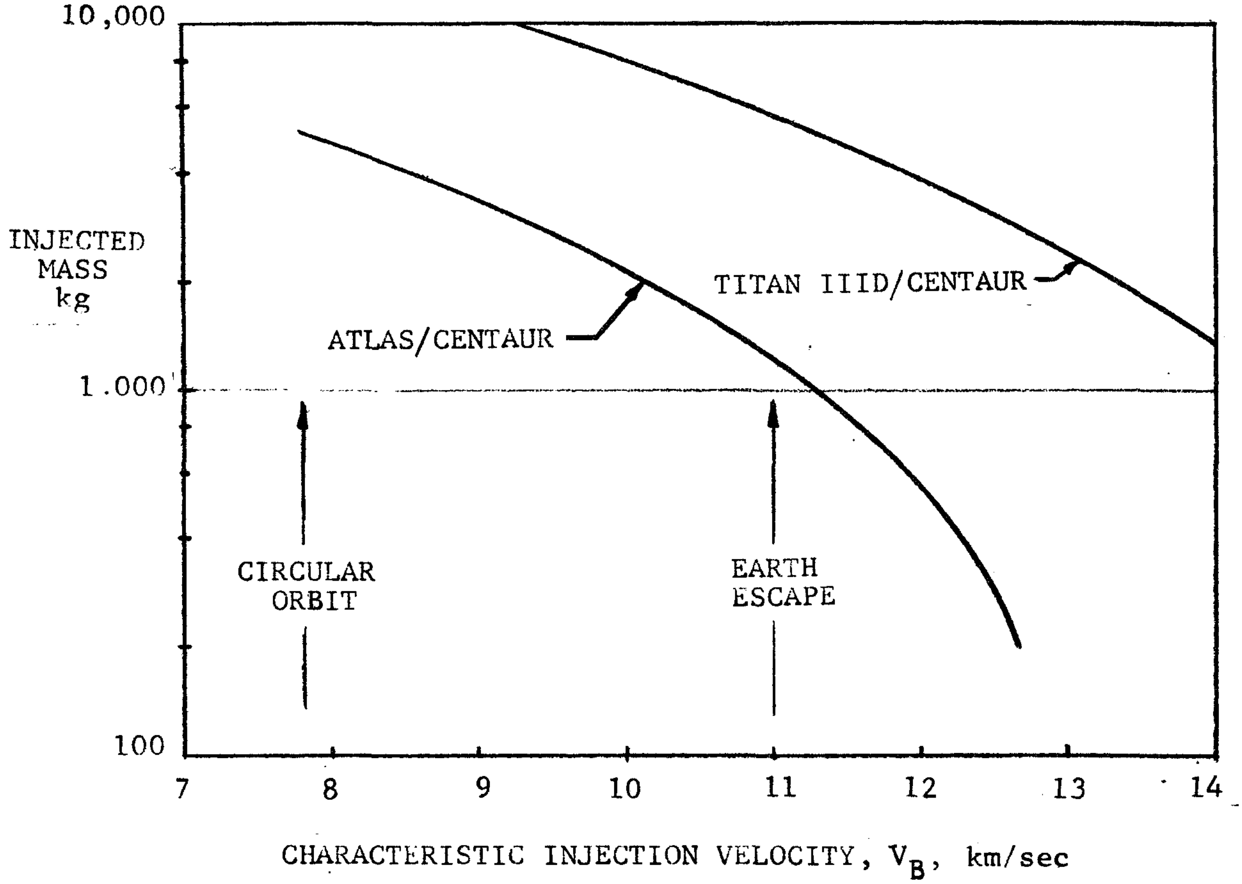

Fig. 1. - INJECTED MASS CAPABILITIES OF TITAN IIID/CENTAUR AND ATLAS/CENTAUR LAUNCH VEHICLES FOR VARIOUS LAUNCH VELOCITIES AT LOW EARTH ALTITUDE

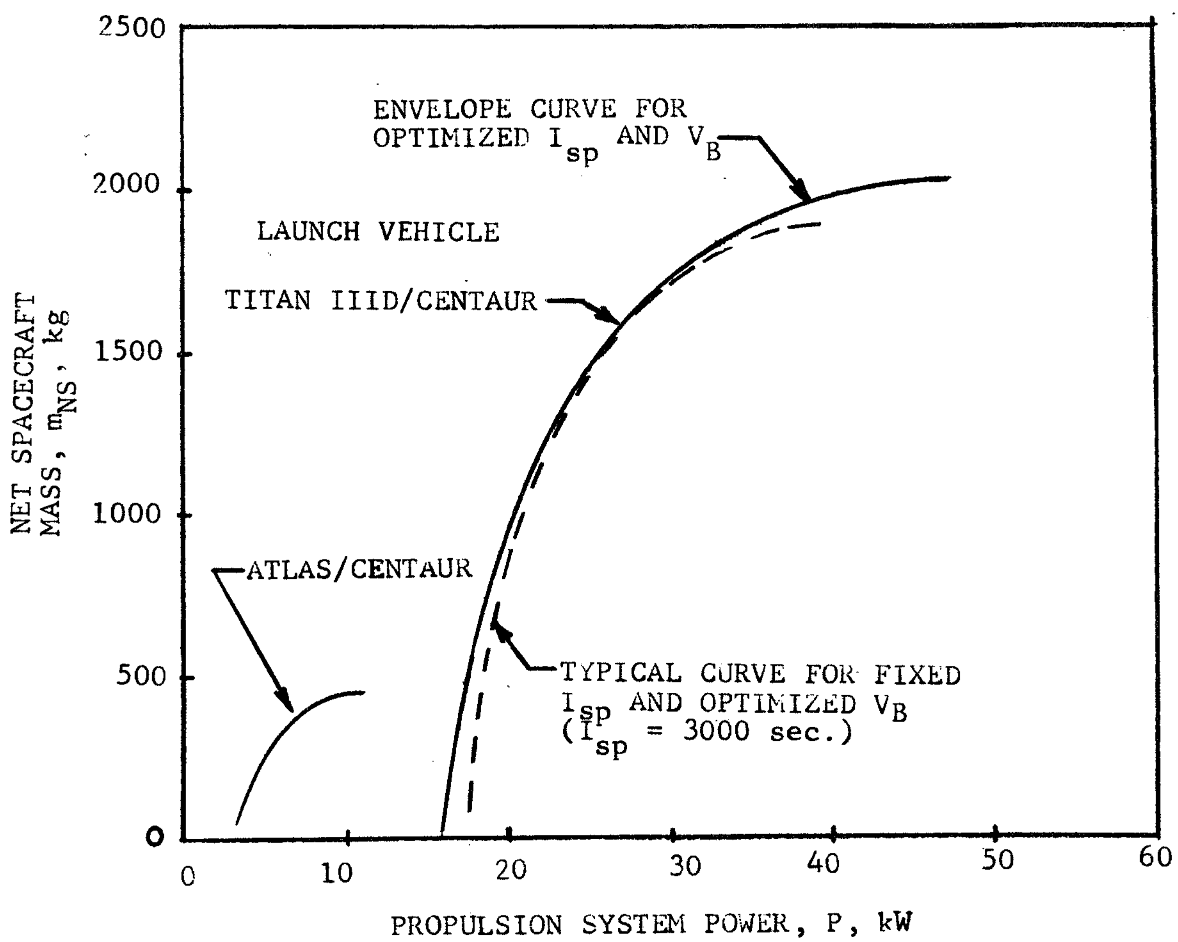

Fig. 2. - MERCURY ORBITER MISSION PERFORMANCE AT SUB-OPTIMUM POWER LEVEL. TRIP TIME, 500 DAYS. SOLAR-ELECTRIC PROPULSION SYSTEM

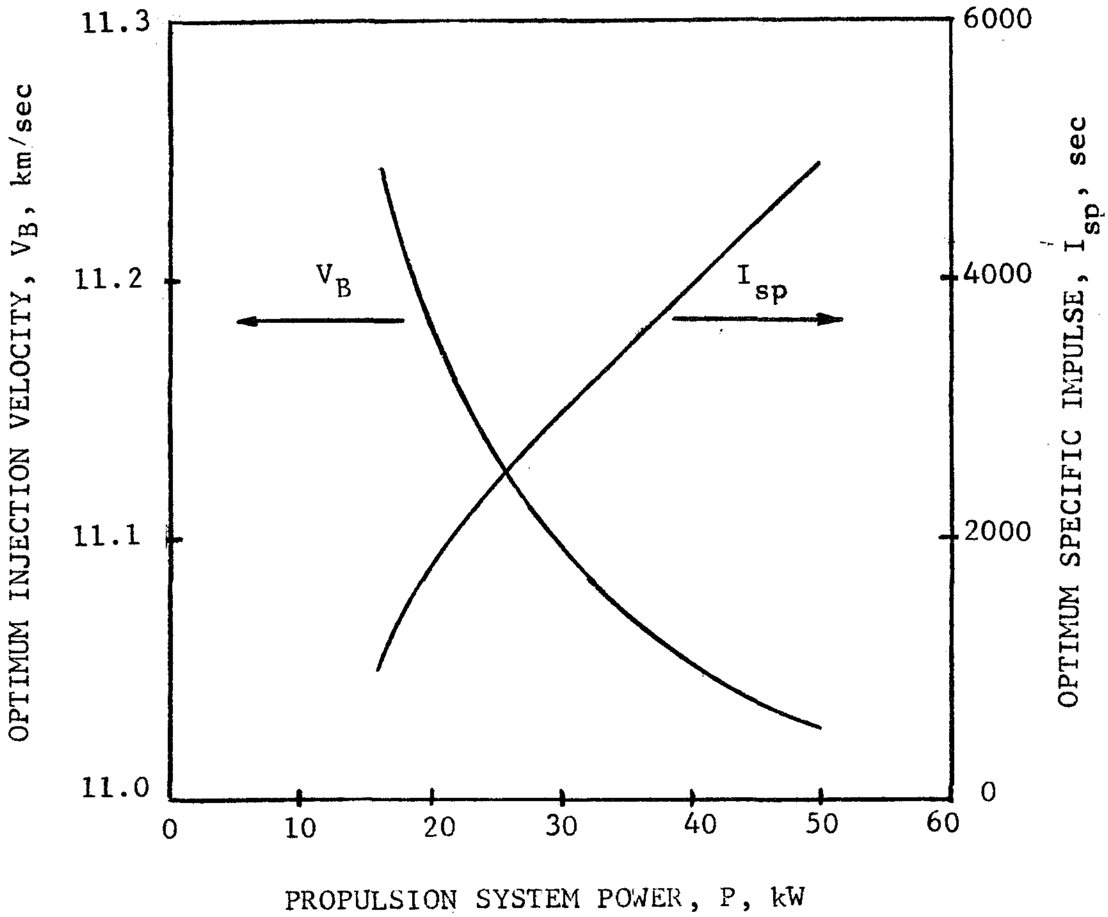

Fig. 3. - MERCURY ORBITER MISSION AT SUB-OPTIMUM POWER LEVEL. TRIP TIME, 500 DAYS. SOLAR-ELECTRIC PROPULSION SYSTEM TITAN IIID/CENTAUR LAUNCH VEHICLE

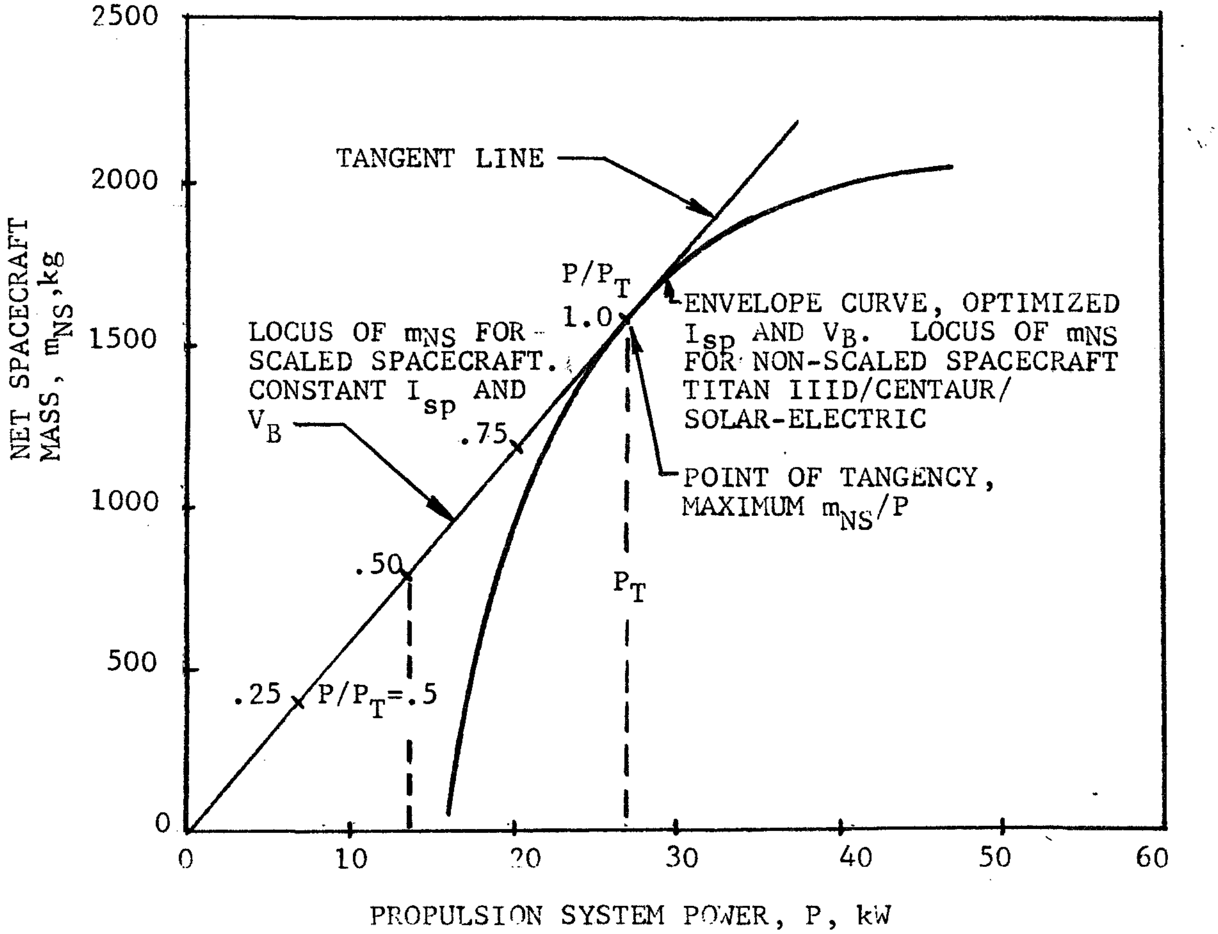

Fig. 4. - SCALING METHOD FOR LOW POWER LEVEL SPACECRAFT. MERCURY ORBITER MISSION WITH 500 DAY TRIP TIME